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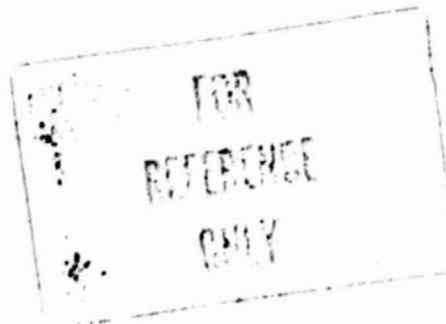
CSC 21H G3/20

Advanced Oxygen-Hydrocarbon Rocket Engine Study

Contract NAS 8-33452
Bi-Monthly Progress Report 33452-M-5
August 1980

Prepared For:
National Aeronautics And Space Administration
George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama 35812

By:
C. J. O'Brien



Aerojet
Liquid Rocket
Company



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ENGINE STUDY

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Prepared by:

C. J. O'Brien

C. J. O'Brien
Project Engineer
ALRC Engineering

Approved by:

J. W. Salmon

J. W. Salmon
Program Manager
ALRC Programs

AEROJET LIQUID ROCKET COMPANY
P. O. Box 13222
Sacramento, California 95813

FOREWORD

This is the fifth bi-monthly progress report submitted for the Advanced Oxygen - Hydrocarbon Rocket Engine Study per the requirements of Contract NAS 8-33452. The work is being performed by the Aerojet Liquid Rocket Company for the NASA-Marshall Space Flight Center. The contract was issued on 15 October 1979. The program inclusive dates for period of performance are 15 October 1979 through 15 February 1981. This report covers the period from 1 June 1980 to 31 July 1980.

The program consists of parametric analysis and design to provide a consistent engine system data base for defining advantages and disadvantages, system performance and operating limits, engine parametric data, and technology requirements for candidate high pressure LO_2 /Hydrocarbon engine systems.

The NASA-MSFC Project Manager is Mr. R. J. Richmond. The ALRC Program Manager is Mr. J. W. Salmon and the Project Engineer is Mr. C. J. O'Brien.

Contributors to this bimonthly report are:

R. Salkeld - Vehicle Trajectory Performance Assessment
H. Mueggenburg - Chamber Design Analysis
R. Ewen - Heat Transfer Analysis

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TABLE OF CONTENTS

	<u>Page</u>
I. Introduction	1
A. Task I - Engine Cycle Configuration Definition	1
B. Task II - Engine Parametric Analysis	2
C. Task III - Engine/Vehicle Trajectory Performance Assessment (Engine Screening)	2
D. Task IV - Baseline Engine Systems Definition	2
E. Task V - Reporting	2
II. Technical Progress Summary	2
A. Task I - Engine Cycle Configuration Definition	4
B. Task II - Engine Parametric Analysis	4
C. Task III - Engine/Vehicle Trajectory Performance Assessment	4
D. Task IV - Baseline Engine System Definition	4
E. Task V - Reporting	19
III. Current Problems	20
IV. Work Planred	20
Task IV	20

LIST OF TABLES

<u>Table No.</u>		<u>Page</u>
I	Engine Cycle G Preliminary Specification	10
II	Engine Cycle I Preliminary Specification	11
III	Baseline Design Heat Transfer Data	12

LIST OF FIGURES

<u>Figure No.</u>		<u>Page</u>
1	Major Milestone Schedule	3
2	JSC Two-Stage Ballistic Launch Vehicle	5
3	Orbital Payload vs Stage 1 Vacuum Isp	6
4	Orbital Payload vs Chamber Pressure for Cycles A Through K (2-Stage)	7
5	LO ₂ /RP-1 Oxidizer-Rich Preburner Staged Combustion Cycle (G) LO ₂ -Cooled	8
6	LCH ₄ Mixed Preburner Staged Combustion Cycle (I) LCH ₄ -Cooled	9
7	LCH ₄ Fuel-Rich Gas Generator Cycle (C) LCH ₄ -Cooled	13
8	Sections of Coolant Channel Layout for LO ₂ /RP-1 Engine	15
9	Conceptual Electroformed Coolant Channel Design	17
10	Conceptual Brazed Coolant Channel Design	17
11	Conceptual Investment Casting Coolant Channel Design	18
12	Conceptual Photoetch Coolant Channel Design	18

I. INTRODUCTION

In the decade of the 1980's and beyond, the nation's expanding space operations may require an improved surface-to-orbit transportation system using advanced booster vehicles which have increased performance and capability compared to the current space shuttle concept. The mixed-mode propulsion principle clearly indicates the potential performance advantages of using high density-impulse rocket propellants in such large ΔV applications. For this reason, hydrocarbon fuels exhibiting increased density relative to liquid hydrogen (LH_2), at the penalty of lower specific impulse, are being considered for the booster propulsion system of space shuttle improvements and derivatives as well as for single-stage-to-orbit and two-stage-to-orbit heavy-payload vehicles.

Preliminary identification and evaluation of promising liquid oxygen/hydrocarbon (LO_2/HC) rocket engine cycles is desirable to produce a consistent and reliable data base for vehicle optimization and design studies, to demonstrate the significance of propulsion system improvements, and to select the critical technology areas necessary to realize such advances.

It is the purpose of this study to generate a consistent engine system data base for defining advantages and disadvantages, system performance and operating limits, engine parametric data, and technology requirements for candidate high pressure LO_2/HC engine systems. The study will also synthesize optimum LO_2/HC engine power cycles and generate representative conceptual engine designs for a specified advanced surface-to-orbit transportation system.

To accomplish the program objectives, the study is composed of four major technical tasks and a reporting task. These tasks and summarized objectives are:

A. TASK I - ENGINE CYCLE CONFIGURATION DEFINITION

Formulate and assess families of high chamber pressure LO_2/HC engine cycles.

I, Introduction (cont.)

B. TASK II - ENGINE PARAMETRIC ANALYSIS

Generate performance, weight, and envelope parametric data for viable concepts based upon historical data and conceptual evaluations.

C. TASK III - ENGINE/VEHICLE TRAJECTORY PERFORMANCE ASSESSMENT (ENGINE SCREENING)

Conduct a preliminary comparison of selected engine cycles utilizing a simplified vehicle trajectory performance model.

D. TASK IV - BASELINE ENGINE SYSTEMS DEFINITION

Prepare preliminary designs of two baseline engine configurations. Conduct heat transfer, turbomachinery, combustion stability, structural, and controls analysis of the baseline engines and components. Conduct a parametric sensitivity analysis including the effects of turbine temperature and number of usable life cycles. Provide the appropriate data in a format suitable for use in vehicle application analyses.

E. TASK V - REPORTING

Provide informal bi-monthly technical and fiscal progress reports, hold program reviews at NASA/MSFC and prepare a final report.

II. TECHNICAL PROGRESS SUMMARY

The overall progress on the program is indicated in Figure 1.

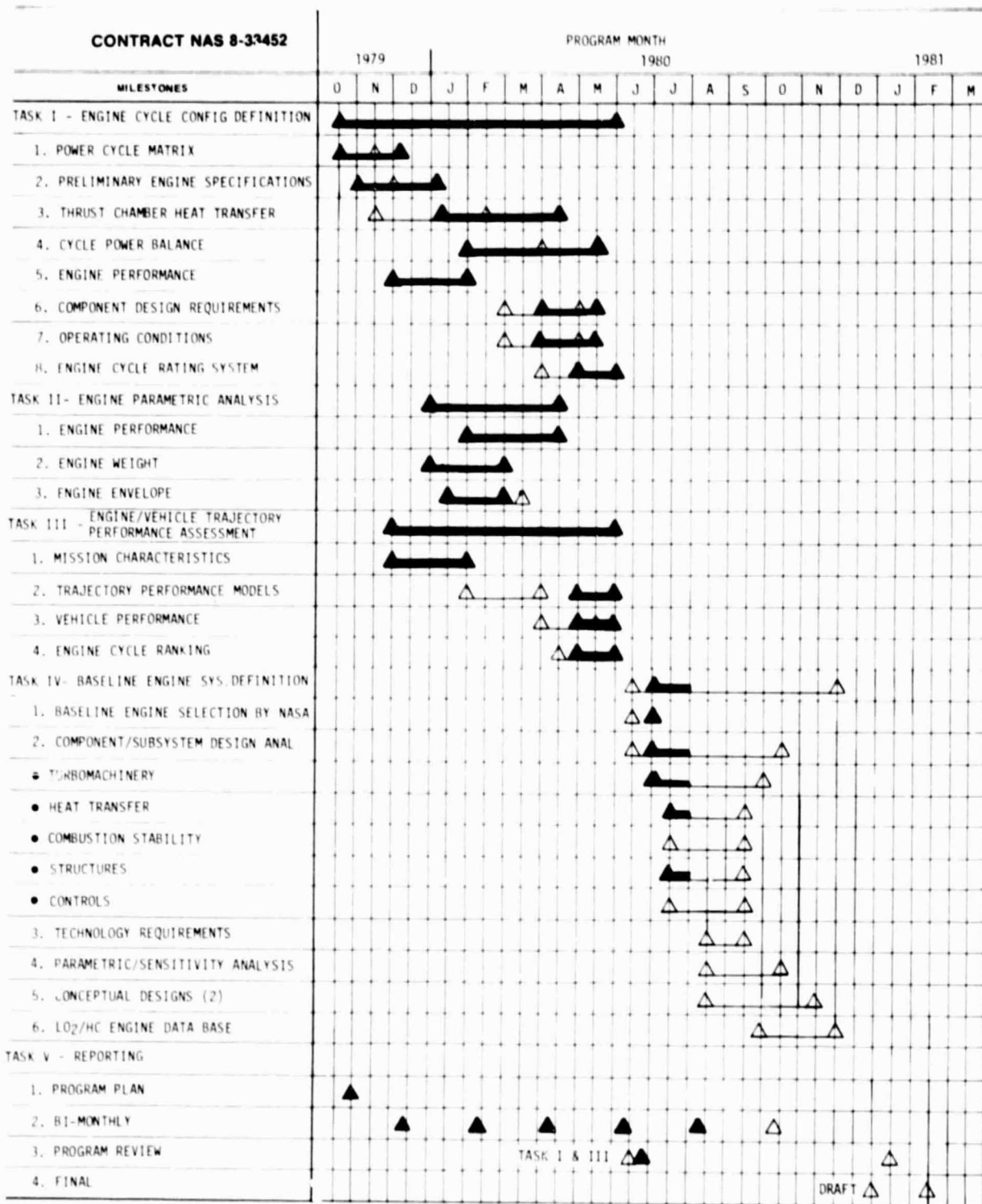


Figure 1. Major Milestone Schedule

II, Technical Progress Summary (cont.)

A. TASK I - ENGINE CYCLE CONFIGURATION DEFINITION

This task is complete.

B. TASK II - ENGINE PARAMETRIC ANALYSIS

This task is complete.

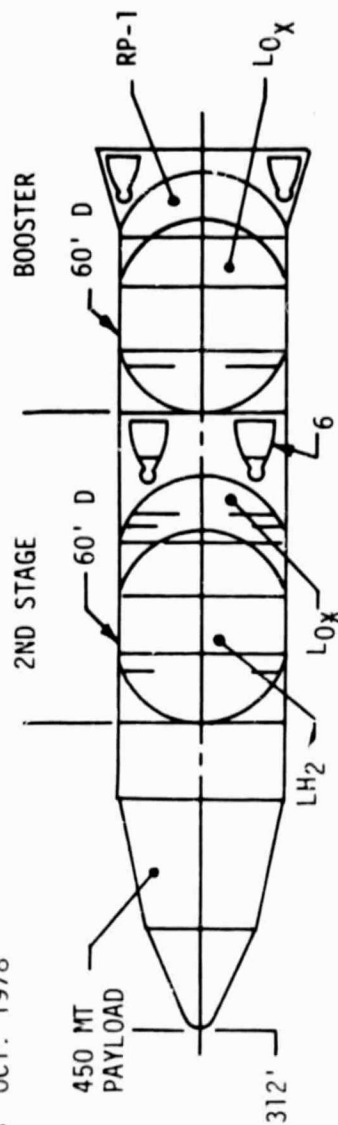
C. TASK III - ENGINE/VEHICLE TRAJECTORY PERFORMANCE ASSESSMENT (ENGINE SCREENING)

This task is complete. Figures 2-4 summarize the final results of this effort for a two-stage ballistic launch vehicle. The variation in delivered payload shown in Figure 4 is partly the result of the chamber pressure capability of each cycle as presented previously in Bi-monthly Report 33452-M-4, 10 June 1980.

D. TASK IV - BASELINE ENGINE SYSTEMS DEFINITION

Two cycles were preliminarily selected for design analysis in Task IV by the NASA Project Manager. They were Cycles G and I as depicted in Figures 5 and 6. The preliminary baseline engine specifications for these cycles are given in Tables I and II. These will be revised slightly to include the heat transfer results from this task, as given in Table III. Subsequent evaluation of the candidate cycles resulted in a revised cycle selection recommendation by the ALRC Program Manager. It has been recommended that Cycle C (methane gas generator) replace Cycle I. This cycle is less complex, and with the consideration of turbine cooling, will attain performance levels nearly as high as the staged combustion cycle. Cycle C is depicted in Figure 7. The cycle choices will be finalized in August.

REF. DOE/ER-0023 OCT. 1978



$O_2/$ RP-1 BOOSTER $O_2/$ PROPANE BOOSTER

PAYLOAD, TONS, 90 x 500 km	454	454
STAGE 1 INERT, TONS	500	485
STAGE 1 PROPELLANT, TONS	4441	4410
STAGE 2 INERT, TONS	233	245
STAGE 2 PROPELLANT, TONS	1937	2056
GROSS LIFT-OFF WEIGHT, TONS	7565	7659
NUMBER OF ENGINES, STAGE 1	12	12
NUMBER OF ENGINES, STAGE 2	6	6
STAGING ALTITUDE, km	43.4	41.3
STAGING VELOCITY (REL), km/sec	1.84	1.70
BOOSTER MAXIMUM DOWN RANGE	381	346

Figure 2. JSC Two-Stage Ballistic Launch Vehicle

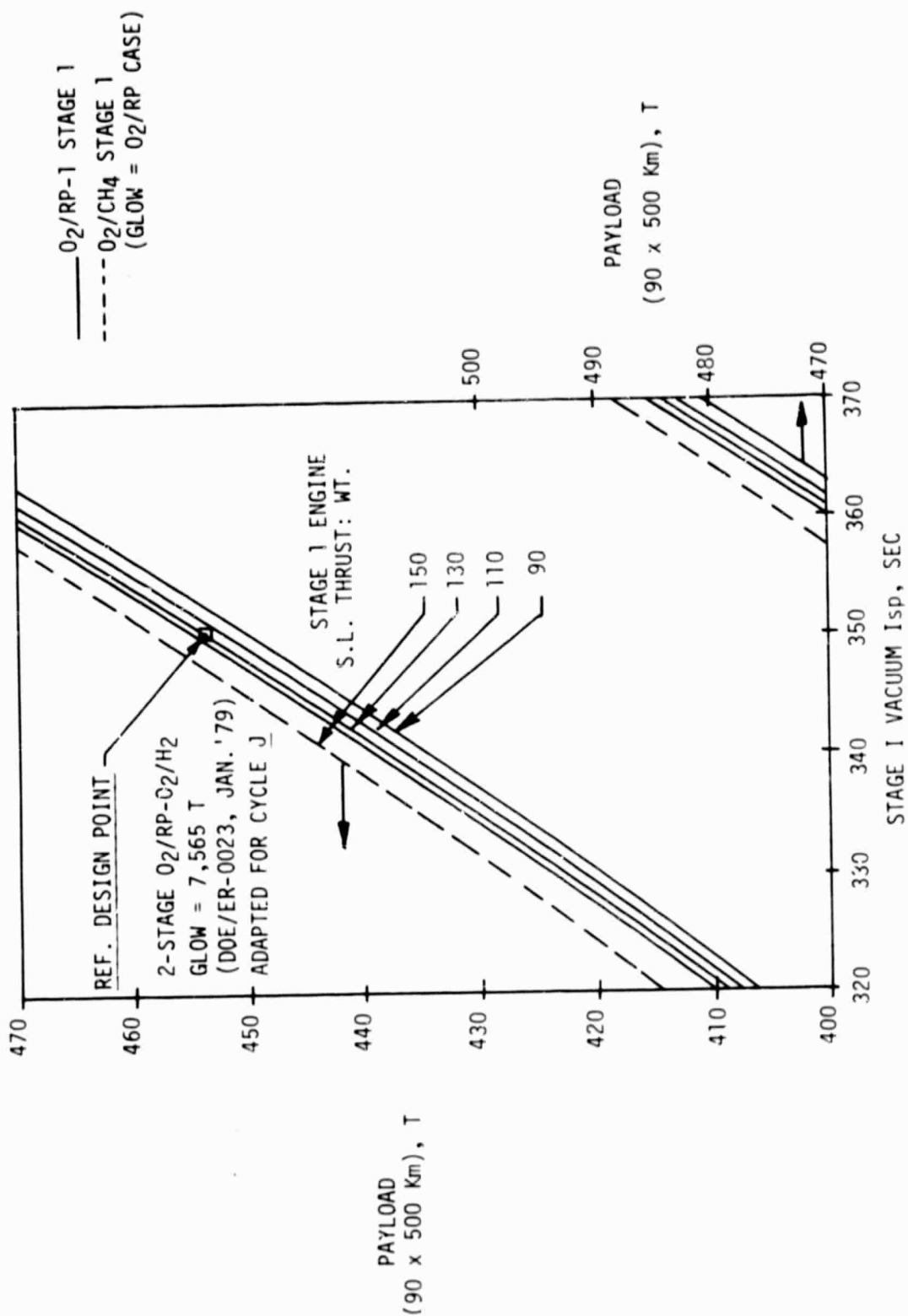
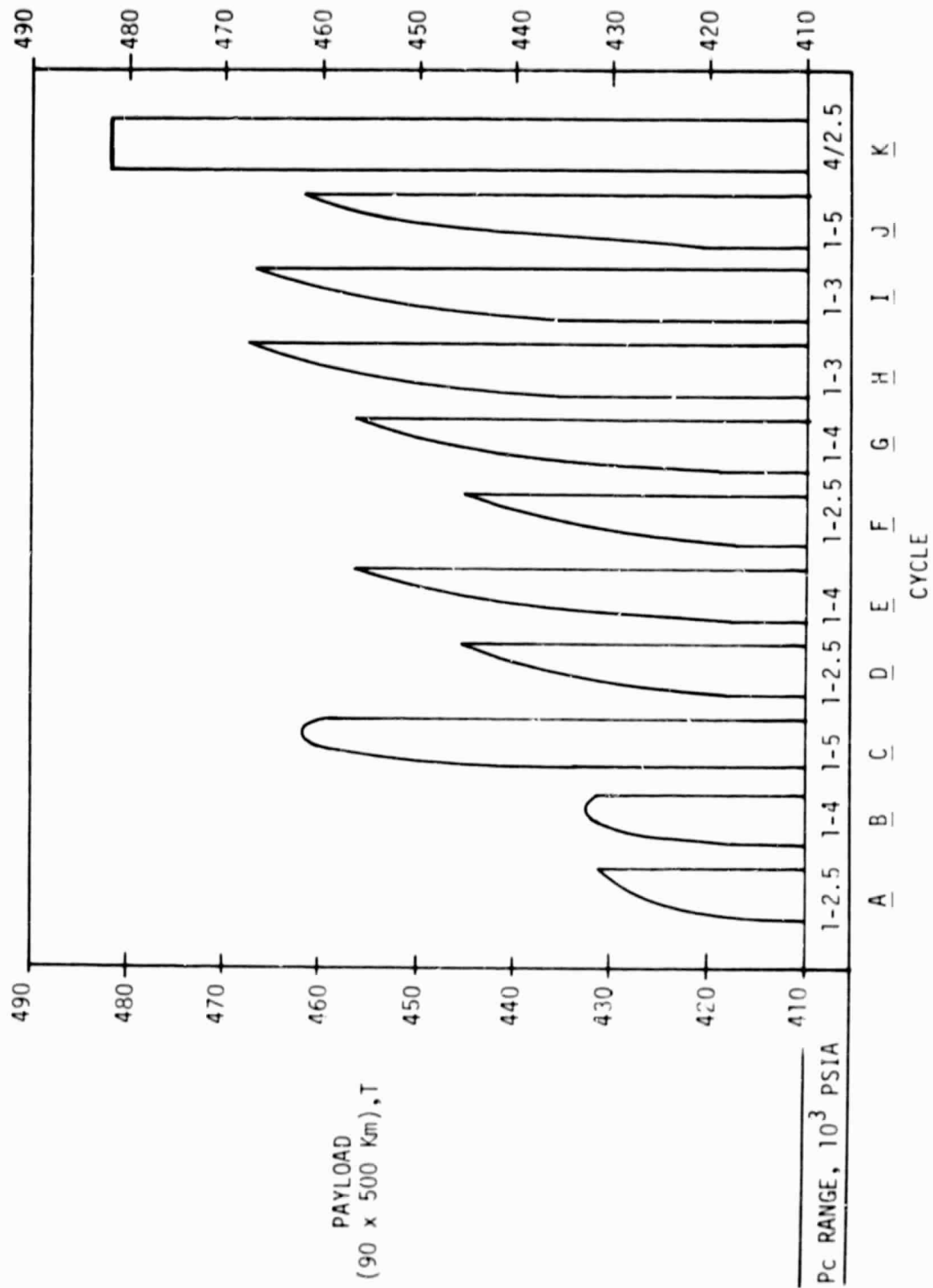


Figure 3. Orbital Payload vs Stage 1 Vacuum Isp



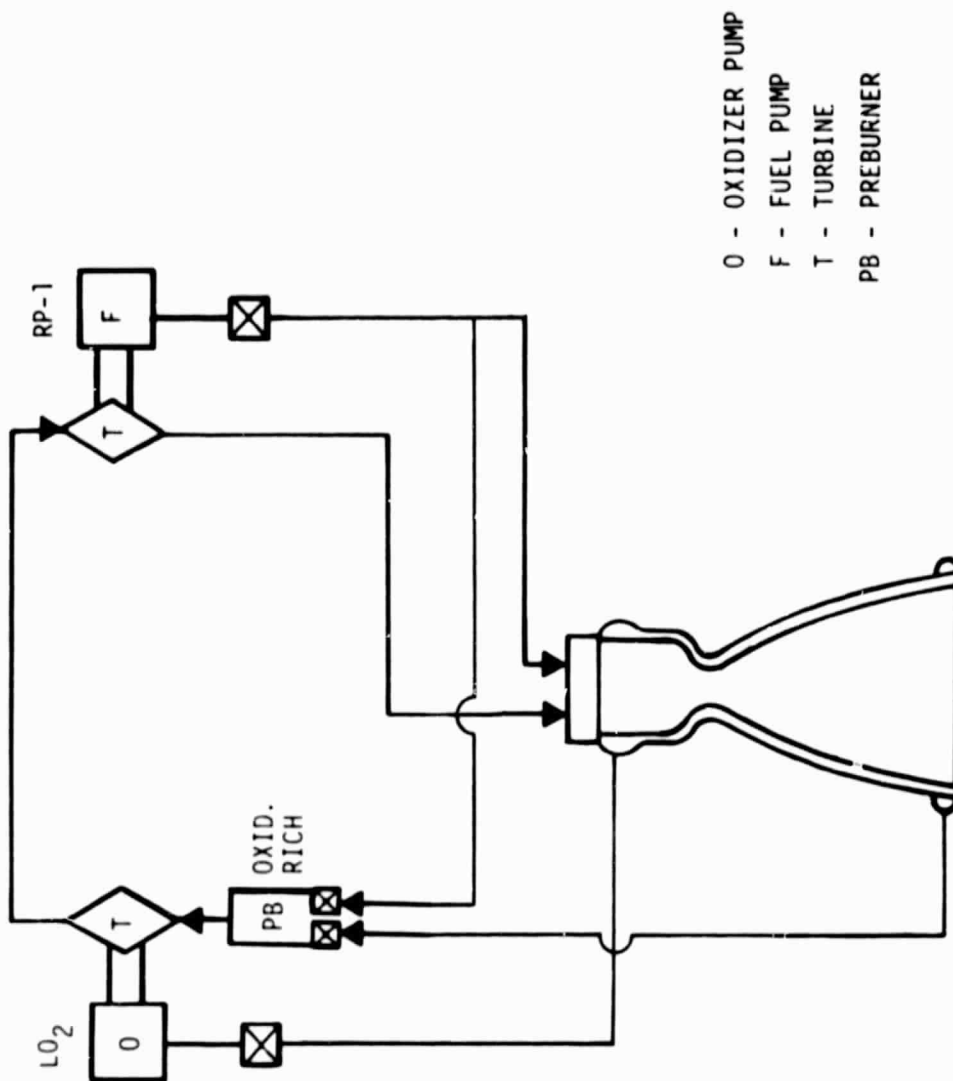


Figure 5. LO₂/FP-1 Oxidizer-Rich Preburner Staged Combustion Cycle (G) LO₂-Cooled

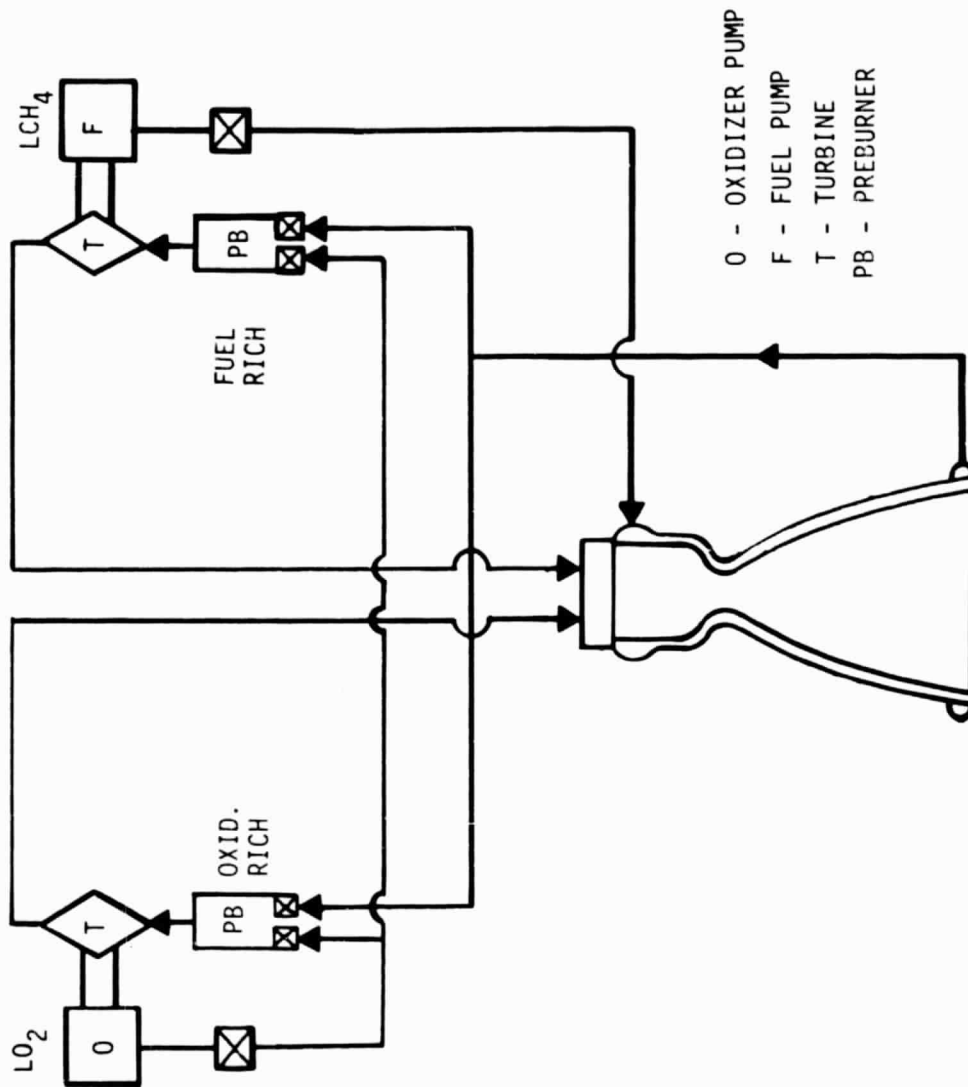


Figure 6. LCH₄ Mixed Preburner Staged Combustion Cycle (I) LCH₄-Cooled

TABLE I
ENGINE CYCLE G PRELIMINARY SPECIFICATION

<u>Propellants</u>	<u>LO₂/RP-1</u>
Chamber Pressure, Psia	3,100
S.L. Engine Thrust, lbF	600,000
Vac. Engine Thrust, lbF	670,832
Mixture Ratio	2.8
Area Ratio	42.5
ODE S.L. Is, Sec	325.9
ODE Vac. Is, Sec	362.8
Is Efficiency, % (V)	0.964
Del. S.L. Is, Sec	312.8
Del. Vac. Is, Sec	349.7
Flowrate, LB/Sec	1918.31
LO ₂ Flowrate, LB/Sec	1413.49
HC Flowrate, LB/Sec	504.82
C*, Ft/Sec	5,927
Throat Area, IN ²	114.0
Exit Area, IN ²	4,845
Exit ODE Pressure, Psia	6.0
PB Mixture Ratio	45
PB LO ₂ Flowrate, LB/Sec	1413.49
PB HC Flowrate, LB/Sec	31.41
Coolant Flowrate, LB/Sec	1413.49
Coolant ΔP, Psi	1,281
Coolant ΔT, °R	77
Turbine Inlet Temp. °R	1,660
Fuel Pump Dischg. P., Psia	3,702
Fuel (PB) Pump Dischg. P., Psia	6,400
LO ₂ Pump Dischg. P., Psia	7,733

TABLE II
ENGINE CYCLE I PRELIMINARY SPECIFICATION

<u>Propellants</u>	<u>LO₂/LCH₄</u>
Chamber Pressure, Psia	3,500
S.L. Engine Thrust, lbF	600,000
Vac. Engine Thrust, lbF	670,381
Mixture Ratio	3.5
Area Ratio	48.0
ODE S.L. Is, Sec	336.2
ODE Vac, Is, Sec	374.1
Is Efficiency, % (V)	.965
Del. S.L. Is. Sec	323.1
Del. Vac. Is, Sec	361.0
Flowrate, LB/Sec	1857.01
LO ₂ Flowrate, LB/Sec	1444.34
HC Flowrate, LB/Sec	412.67
C*, Ft/Sec	6,098
Throat Area, IN ²	100.6
Exit Area, IN ²	4,827
Exit ODE Pressure, Psia	6.0
PB Mixture Ratio	.39/41.5
PB LO ₂ Flowrate, LB/Sec	149.59/1294.75
PB HC Flowrate, LB/Sec	383.58/29.10
Coolant Flowrate, LB/Sec	383.58
Coolant ΔP, Psi	1,370
Coolant ΔT, °R	150
Turbine Inlet Temp. °R	1860/1660
Fuel Pump Dischg. P., Psia	8,272
LO ₂ (FPB) Pump Dischg. P., Psia	6,847
LO ₂ Pump Disch. P., Psia	6,654
Fuel (OPB) Pump Dischg. P., Psia	6,504

TABLE III
BASELINE DESIGN HEAT TRANSFER DATA

<u>Cycle</u>	<u>Propellants</u>	<u>Coolant</u>	<u>Chamber Pressure (Psia)</u>	<u>Coolant F = 600K (Psi)</u>	<u>Pressure Drop F = 1M lbF (Psi)</u>
G	LO ₂ /RP-1	LO ₂	3100	1250	1532
I	LO ₂ /LCH ₄	LCH ₄	3500	1350	1535

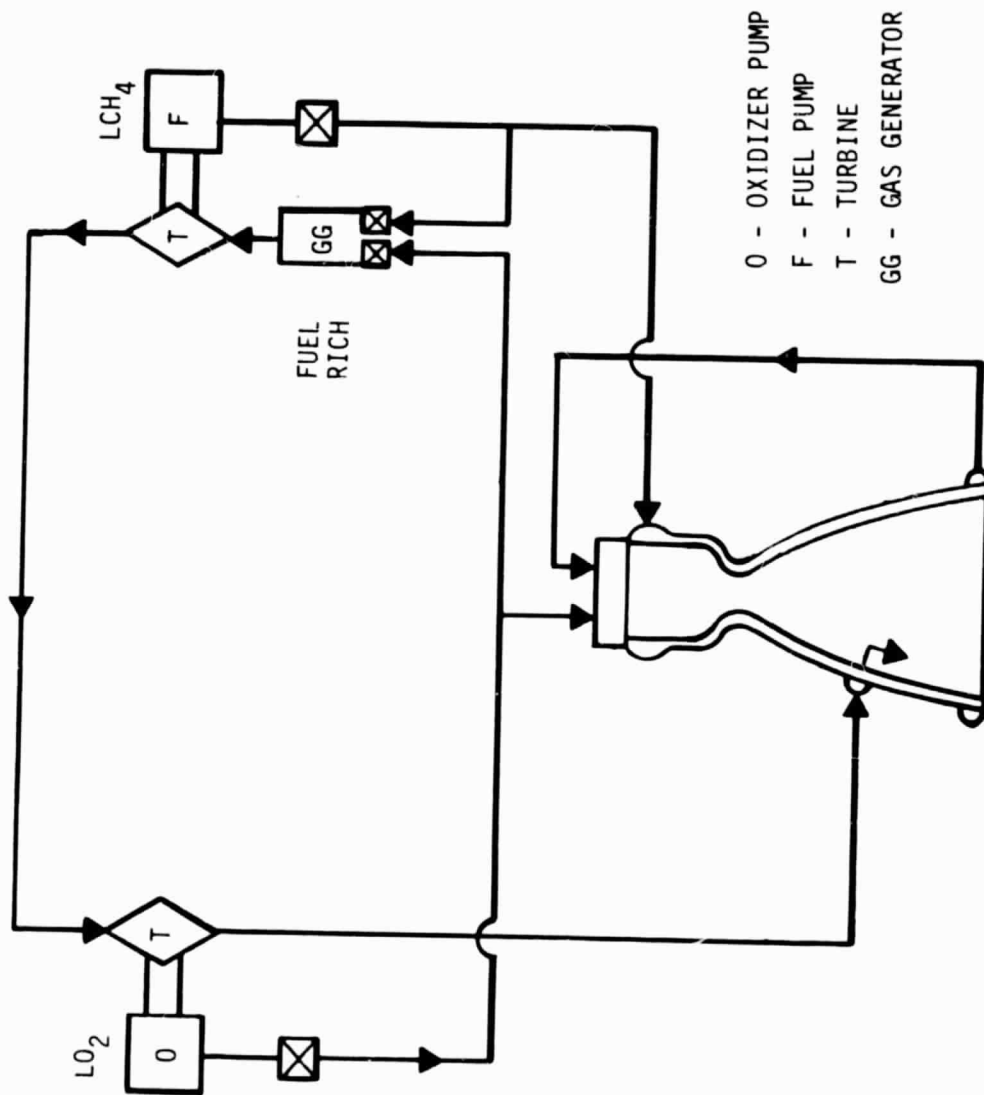


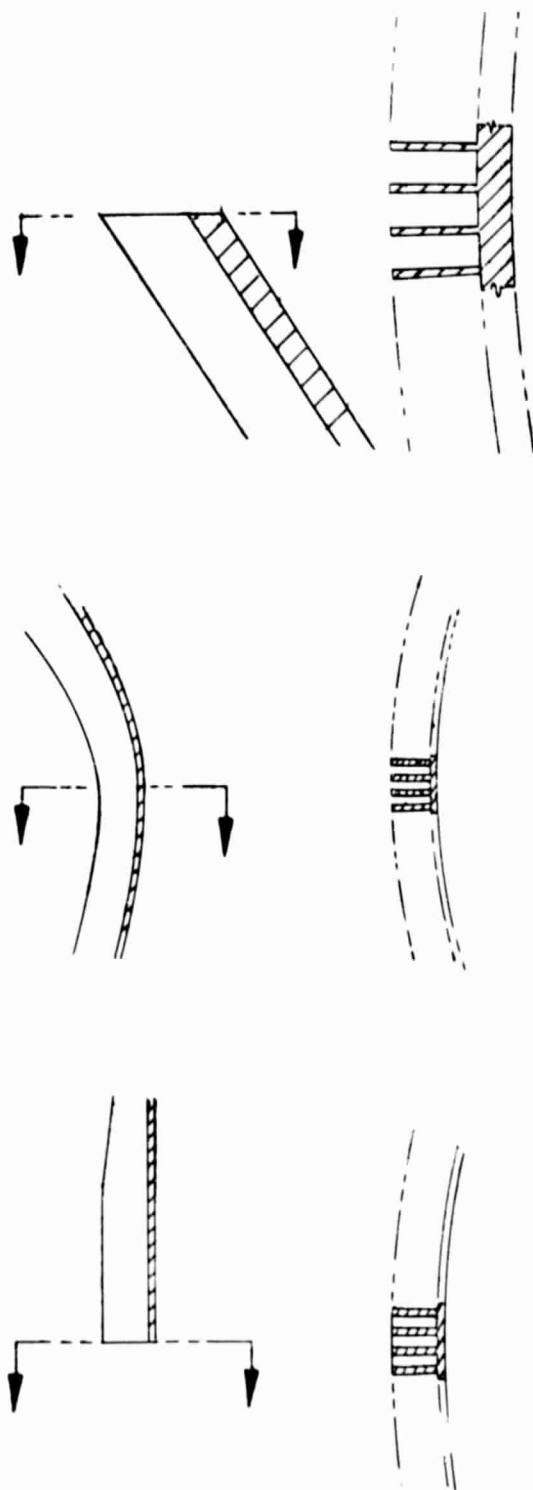
Figure 7. LCH₄ Fuel-Rich Gas Generator Cycle (C) LCH₄-Cooled

II, D, Task IV - Baseline Engine Systems Definition (cont.)

Preliminary design analysis of the heat transfer subsystem was performed to establish major technology requirements. Chamber coolant slot layouts for two LO_2 /RP-1 engines were prepared. Both engines are of a 600,000 lbf thrust level, utilize LO_2 cooling, and operate at either 3,000 or 4,000 psia chamber pressure. Figure 8 illustrates typical sections of the slot layout for the 4,000 psia chamber.

Coolant channel fabrication feasibility was checked by considering state-of-art approaches as well as advanced manufacturing processes. The design can be manufactured conventionally, i.e., with a slotted zirconium copper chamber with an electroformed nickel closure similar to the OMS chamber. However, the cost of the slotting operation of the chamber will not only be proportionately greater than the OMS because of the size difference but also because of two significant channel parameter differences. The greatest cost impact is the 0.66 inch maximum depth of channel as compared to the 0.16 inch on the OMS chamber. Not only will a greater diameter slitting saw be required but at the two chamber extremes, where coolant enters and leaves the chamber, the slots will have to be deepened locally. This is required because the larger radius cut leaves a greater chamber wall thickness. Deepening the channels locally will probably have to be performed with the more expensive Electrical Discharge Machining (EDM) process. The second cost impact is the very narrow but constant channel wall land width of .04 inch from the throat to the aft end of the chamber. It may be possible to redesign the aft end of the chamber to avoid this narrow land. Constant width wall lands are normally machined by straddle milling but it is very doubtful that a 0.04 inch wall can be machined to a depth of over 0.6 inch. For this reason it is more likely that for very narrow lands every other channel will be cut, then filled with Rigidax prior to machining the remaining channels.

THRUST 600,000 lbf
CHAMBER PRESSURE 4,000 psia
COOLANT LO_2



$D_c = 16.048$

CHAMBER
SECTION

$D_t = 10.553$

THROAT
SECTION

$D_e = 30.003$

NOZZLE EXIT
SECTION

Figure 8. Sections of Coolant Channel Layout for $\text{LO}_2/\text{RP-1}$ Engine

II, D, Task IV - Baseline Engine Systems Definition (cont.)

Alternate fabrication concepts considered for these advanced engine cooling system designs are shown in Figures 9 through 12. The first concept shown in Figure 9 shows the cross section of an all electroformed chamber configuration. In this concept, individual tubes are first electroformed around a wax preform simulating the flow channel. These tubes are then assembled onto a mandrel forming every other coolant passage. The vacant spaces between the tubes are then filled with wax permitting a closeout shell of electroformed nickel to be formed. The chamber mandrel is then removed permitting the copper liner to be electroformed to the inside thus completing the all electroformed assembly.

A second concept is shown in Figure 10. In this concept preformed U-tubes are brazed to the copper liner forming every other coolant passage. The vacant spaces between the U-tubes are filled with wax prior to electroforming the nickel close-out structure.

The third alternate fabrication concept is shown in Figure 11. Individual copper ribs are manufactured by either the investment casting process or by swedge forming to produce an optimum heat transfer configuration fin. These preformed copper ribs are then assembled on a mandrel to form the coolant channel circuit as shown. The electroformed nickel closure is then deposited followed by electroforming the copper liner.

The fourth alternate fabrication concept is shown in Figure 12. In this concept the chamber ribs are fabricated by the photoetch process. The 0.04 inch thick through etched rib edges are squared off on a drum sander and assembled into the forward and aft flange which serves as the fixture to achieve proper radial and circumferential alignment. The vacant spaces or flow passages between the ribs are filled with wax permitting both the closure wall and liner wall to be electroformed. The photoetched ribs could also be brazed into a premachined copper liner requiring only the closure wall to be electroformed.

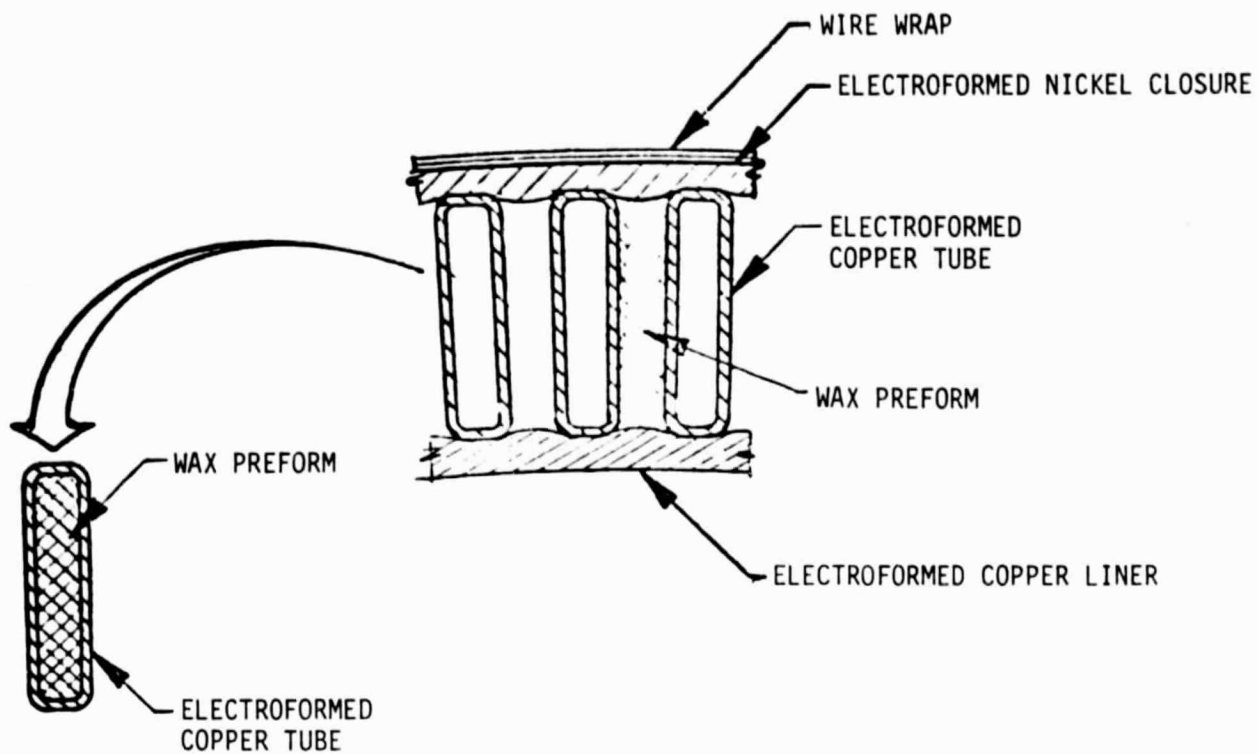


Figure 9. Conceptual Electroformed Coolant Channel Design

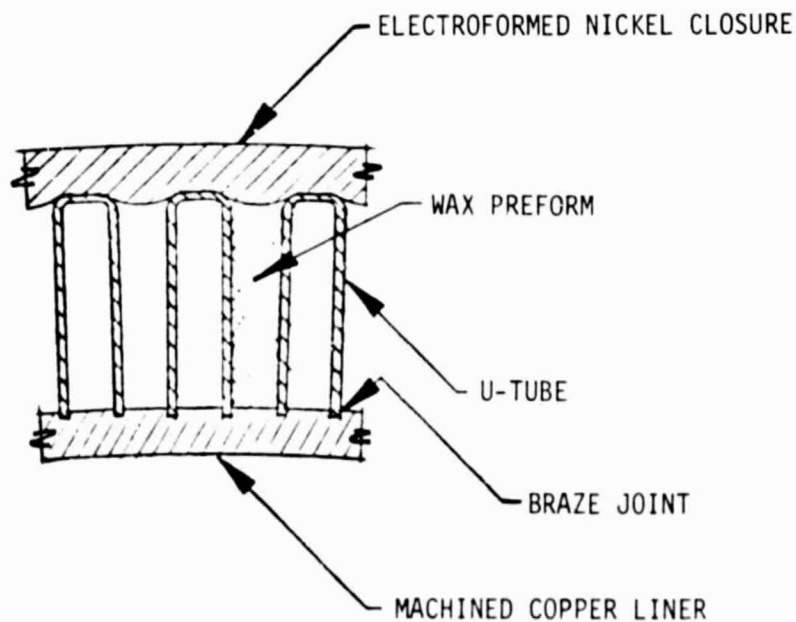


Figure 10. Conceptual Brazed Coolant Channel Design

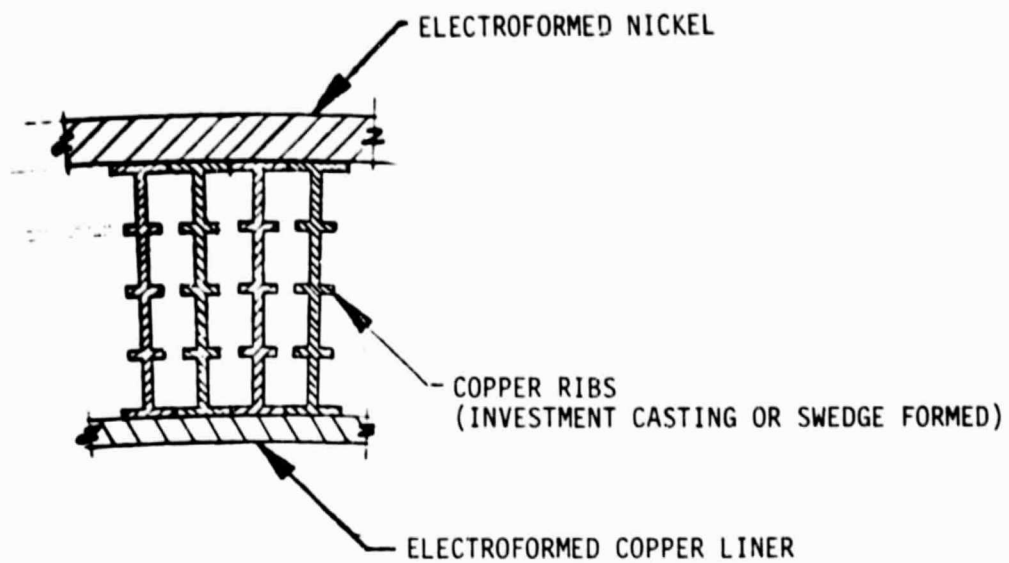


Figure 11. Conceptual Investment Casting Coolant Channel Design

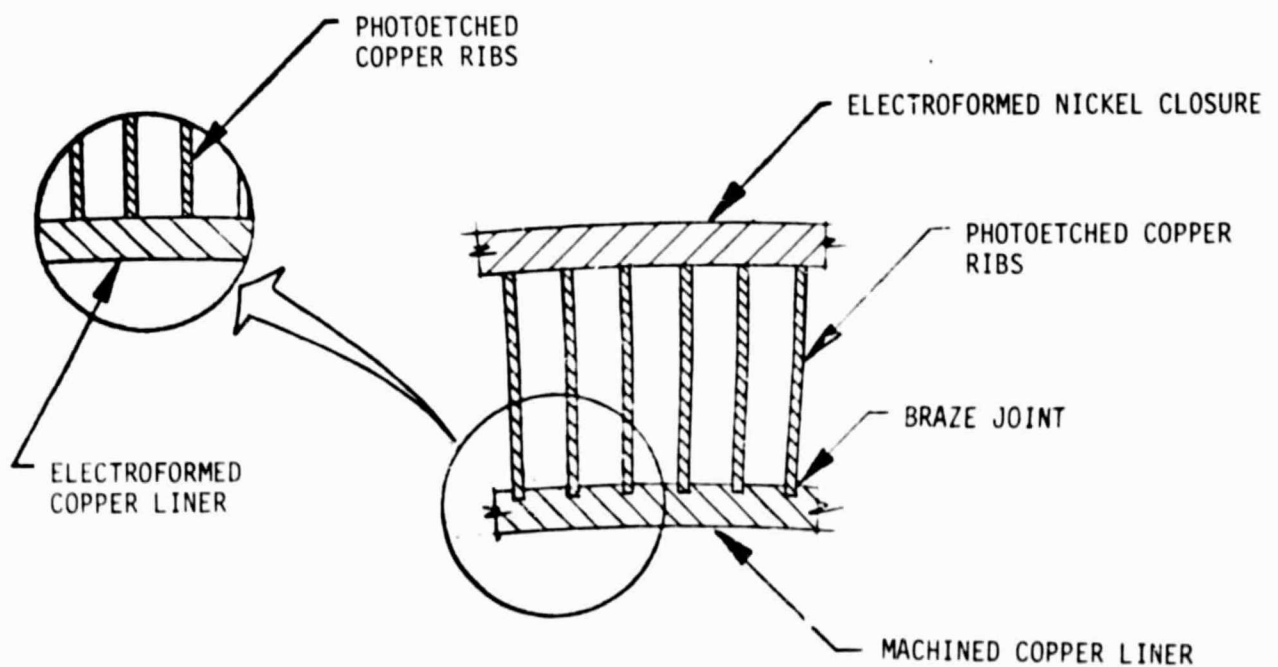


Figure 12. Conceptual Photoetch Coolant Channel Design

II, D, Task IV - Baseline Engine Systems Definition (cont.)

Before a final design can be recommended and selected the overall chamber cooling concept configuration must be analyzed in more depth. For instance, a feature which may eliminate excessive channel depth is to put the inlet torus just downstream of the throat, i.e. at an $A_e/A_t=2$. This would also increase the rib width from 0.04 inch to a more acceptable width required not only for ease of manufacturing, but for structural adequacy. Another approach to be considered would be to segment the coolant circuit of the chamber into four or more axial sections, each having its own inlet and outlet torus, each flowing only a portion of the available cooling. This scheme would reduce the required coolant channel cross sectional area (reduced channel depth) and would be structurally superior because of the increased number of tori which would act as hoop bands around the high pressure chamber.

E. TASK V - REPORTING

A review of the Tasks I - III results was accomplished at MSFC on 26 June. Discussions were held concerning the various engine cycles and the merits of including LO_2/LC_3H_8 parametric engine data in the program results. The following agreements were tentatively made:

(1) Cycle G: $LO_2/RP-1$, LO_2 -cooled, LO_2 -rich preburner, staged combustion cycle was selected for Task IV design analysis.

(2) Cycle I: LO_2/LCH_4 , LCH_4 -cooled, LCH_4 -rich and LO_2 -rich preburners, staged combustion cycle was selected for Task IV design analysis. This recommendation has been changed to LO_2/LCH_4 , LCH_4 -cooled LCH_4 -rich gas generator, as previously indicated.

II, E, Task V - Reporting (cont.)

(3) No turbomachinery design effort is to be conducted in Task IV. A design analysis of turbomachinery parameters and efficiencies will be made to establish technology requirements with and without high temperature turbines.

(4) $\text{LO}_2/\text{LC}_3\text{H}_8$ parametric engine data for Cycles G, I and J will be generated similar to that obtained in Tasks I and II. This added scope can be accomplished through reduction in scope of the more detailed preliminary design tasks (turbomachinery, controls and design).

III. CURRENT PROBLEMS

There is presently a two-week slip in the program as scheduled because of the lateness of the Task review. It is anticipated that this slip will not cause a slip in the completion date of the program.

IV. WORK PLANNED

TASK IV

Conduct $\text{LO}_2/\text{LC}_3\text{H}_8$ heat transfer study with LC_3H_8 (propane) as coolant. Conduct turbomachinery analysis, combustion stability, controls and structures subtasks. Prepare engine cycle balances for $\text{LO}_2/\text{LC}_3\text{H}_8$ engine Cycles G, I and J.